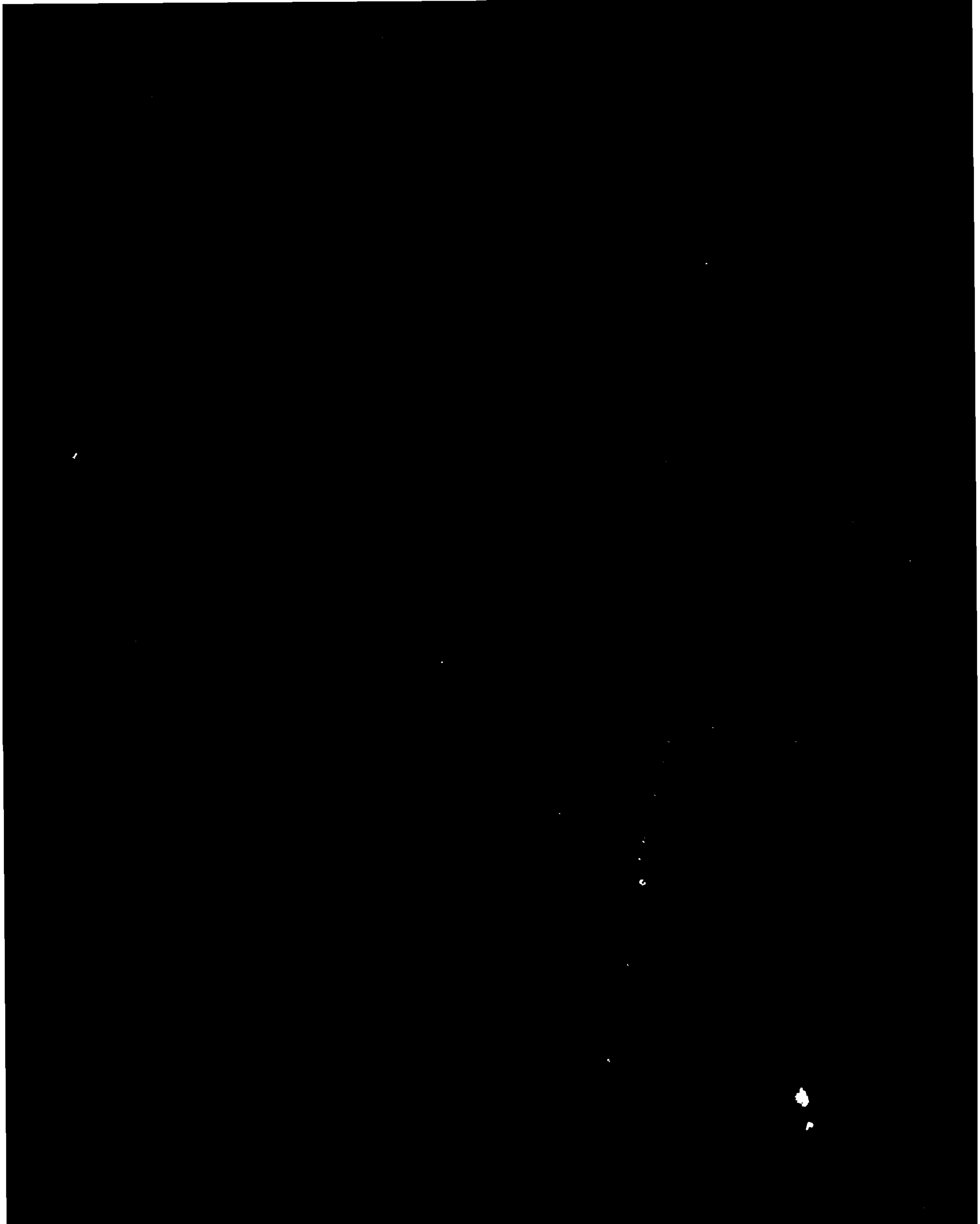
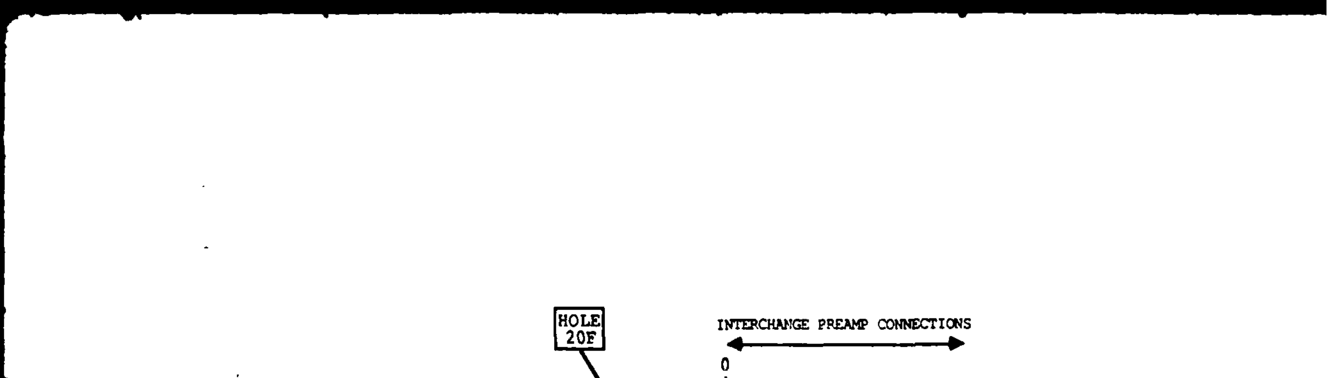
AD-A186 714 LABORATORY STUDIES RELATED TO IN-FLIGHT ACOUSTIC EMISSION MONITORING(U) AERONAUTICAL RESEARCH LABS MELBOURNE (AUSTRALIA) S R LAMB APR 87 ARL-MAT-TM-393

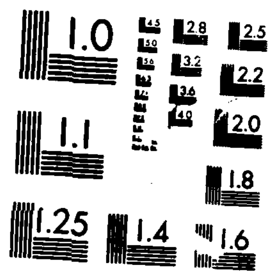
1/1

UNCLASSIFIED DODA-AR-004-531 F/G 20/1 NL

END




HOLE
20F
INTERCHANGE PREAMP CONNECTIONS
0


1.0
1.1
1.25
1.4
1.6
1.8
2.0
2.2
2.5
2.8
3.2
3.6
4.0

(12)

ARL-MAT-TM-393

AR-004-531

AUSTRALIA

DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORIES

MELBOURNE, VICTORIA

Aircraft Materials Technical Memorandum 393

LABORATORY STUDIES RELATED TO IN-FLIGHT ACOUSTIC EMISSION MONITORING (U)

by

S.R. Lamb

APRIL 1987

AR-004-531

DEPARTMENT OF DEFENCE
DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION
AERONAUTICAL RESEARCH LABORATORIES

Aircraft Materials Technical Memorandum 393

LABORATORY STUDIES RELATED TO IN-FLIGHT ACOUSTIC EMISSION MONITORING (U)

by

S.R.Lamb

SUMMARY

Programmed load testing of a structural member from a MACCHI aircraft was undertaken in an attempt to determine the source of the acoustic emission previously recorded during in-flight monitoring of the same component and to compare results obtained from different equipment. Although crack growth during laboratory testing appeared similar to in-flight crack growth, the pattern of recorded AE was markedly different. The laboratory tests are described and an explanation of the test results sought.

POSTAL ADDRESS: Director, Aeronautical Research Laboratories,
P.O. Box 4331, Melbourne, Victoria, 3001, Australia.

CONTENTS

1. INTRODUCTION

The use of high-strength structural members in modern military aircraft has led to a need for major improvements in Non Destructive Inspection (NDI) capability. This need has been met to a large extent by improvements in existing NDI procedures and the introduction of some new techniques such as magnetic rubber inspection (MRI). The cracks which are of concern, however, are usually small - up to a few millimetres in depth - and the inspection of major structural components is usually made difficult by the complexity of the structure and its relative inaccessibility.

As a result, there is increasing interest in NDI techniques which can be applied in-situ, and even in-flight; to this end, there is considerable interest in the use of Acoustic Emission (AE) methods, and the work described in this report forms part of a more extensive program which investigated some aspects of the use of AE for in-flight monitoring, by collecting AE and crack growth data from a major structural component in an RAAF MACCHI jet trainer aircraft.

In-flight tests on a MACCHI aircraft demonstrated that acoustic emission (AE) can be detected in the flying environment [1,2]. Laboratory tests were undertaken in an attempt to identify the source of the recorded AE and to compare results obtained from different equipment.

2. TEST PROGRAM

The AE system used on the aircraft comprised two transducers bonded to the steel tension member of the centre plane assembly (wing carry- through structure), and connected through preamplifiers and filters to a processing and recording (EPROM) unit. A zone-isolation method was used to reduce the effects of extraneous structure-borne noise and to ensure that recorded signals came only from the regions of interest (viz. holes 20F and 20R, Figure 1)

The AE counts* from the region of the cracked holes were progressively summed over a total of 838 flights, representing approximately 1000 flight hours, and plotted against the crack surface length (Figure 2). Linear regression analysis provided a line of best fit with a correlation coefficient of 0.99.

When the cracking in the centre plane tension boom reached a pre-determined maximum allowable size, the component was removed from the aircraft and prepared for further cycling in a laboratory test rig; AE and crack length monitoring was continued at the same locations.

---

* Identified as 'VALID AE' in references 1,2 and 3

In the laboratory test the component was subjected to the repeated application of a spectrum of loads representative of the previous flight history; these loads were applied in-programs simulating a sequence of 200 flights (the average duration of each simulated flight being about one hour). The highest positive (tensile) load applied during testing was equivalent to a 6.0g aircraft manoeuvre; the highest negative (compressive) load corresponded to a -1.563g manoeuvre.

To facilitate fractographic analysis of the fatigue crack surface, a short series of loads, which would leave an identifiable mark (over a short distance) on the fatigue crack surface, was applied at the beginning of the laboratory test. Reference to the 'g'-meter records from the aircraft, and to in-flight crack propagation data obtained from magnetic rubber (MRI) replica records, indicated that 150 repeated applications of a 3.5g load should produce a distinctive mark on the crack surface. This formed the so-called 'marker' loads sequence.

3. ARRANGEMENT OF COMPONENT IN TEST RIG

In the laboratory test, the component had all fasteners removed and was free of all normal fittings, attachments, and other contacts which could have generated AE in the aircraft. The only connections through which mechanical noise of extraneous origin could enter the component were from:

(i) the cross-heads of the hydraulic fatigue testing machine; and

(ii) the centre support framework (labelled A in Figure 3) linking the specimen to the frame of the testing machine, to limit bending loads during compressive loading.

Direct contact between the constraining device and the specimen was prevented by the use of teflon sheet. Associated bolts were coated with a lead-base anti-fretting compound. Plasticene was used on the cross-heads and the attachment lugs of the spar to attenuate any surface waves originating in the mechanical and hydraulic elements of the test rig. All electrical cables (other than those necessary for AE monitoring) were kept well away from the specimen and the monitoring transducers.

The in-flight AE system was used in the laboratory test; the transducers and wiring were permanently attached to the spar for the in-flight tests and hence were not changed. A second system, of a different kind (AET Corp. Model 3000-LOCATOR), was also fitted for the laboratory test. The LOCATOR displays the activity and location of AE sources along a line joining its two transducers, and was used to monitor the component over practically its full length. Transducers and preamplifiers/filters for the LOCATOR system were carefully selected to match, as closely as possible, the characteristics of the in-flight AE system.

4. TEST PROCEDURE

The position and extent of the isolation zone were set in the laboratory test to be the same as in the flight trials (see Figure 1), and the threshold voltage level remained unchanged. A pulser was used to confirm the extent of the isolation zone. For the application of the 'marker' loads, the up-date time for the EPROM was set so that the effect of individual load cycles could be recorded.

Programmed fatigue loading of the disassembled component was continued (after the application of the 'marker' loads sequence) with crack growth monitoring, for a total of 9 programs, corresponding to 1800 simulated flights*. Continuous AE recording from the region of the cracked holes was undertaken with the aircraft AE system throughout this testing period. Also, photographs were taken of the AET 3000 CRO screen after each loading program (corresponding to 200 simulated flights). The surface lengths of the cracks in the subject holes were measured at the end of the application of each block of 200 flights (i.e. at the end of each program) using the magnetic rubber technique.

5. TEST RESULTS

The 'marker' loads applied to the test component produced a surface crack extension of 0.25mm, during which the total number of AE counts recorded was 18. The progressive number of counts, from the start of the in-flight monitoring to the completion of the laboratory test, are plotted against crack surface length in Figure 4.

The rate of emission in the laboratory test remained relatively low for the first 1000 simulated flights, represented by programs 1 to 5 inclusive. In this period, the surface length of the crack increased by 2.28 mm and 1260 emissions were recorded. Previously, in the flight trials, 3.76 mm of crack extension, over 838 actual flights, was accompanied by 16344 emissions from the same zone. Subsequent to laboratory program 5, 20743 emission were recorded during 800 simulated flights, with 1.04 mm increase in crack surface length. These results are summarised in Table 1.

---

* A total of 10.5 programs, equivalent to 2100 aircraft flights, have in fact been applied to the test component so far. However, because the cracks at hole 20F grew very rapidly after the completion of program 9 and broke through the bottom edge of the spar flange, the experimental data obtained subsequent to program 9 was not considered to be relevant to the study of the more realistic smaller cracks.

Photographic records of the AET 3000-LOCATOR CRO screen taken at the end of several of the laboratory test loading programs, are shown in Figure 5. In each case the location of the zone of interest, containing the cracked holes, is indicated on the horizontal scale. Of particular interest is the marked increase in AE activity in the specified region recorded by the system after program 5, confirming the record of the aircraft monitoring system for the same zone (Figure 4) over the same period of laboratory testing. The noise 'peaks' at approximately the centre of each of the photographic records of Figure 5 have been generated by the centre support framework, labelled A in Figure 3.

Upon completion of the programmed loading, the test component was closely re-examined. Evidence (Figure 6) was found of heavy surface damage on the tension boom close to the cracked zone (but outside the isolation zone). This damage was caused by rubbing and/or fretting between the top surface of the boom and the surface of another structural member of the centre-plane assembly. This must have occurred when the component was in the aircraft.

No evidence was found of significant fastener/component surface interaction (rubbing, fretting) in (or at) either of the cracked holes within the isolation zone.

6. POSSIBLE SOURCES OF AE DURING FLIGHT

Possible sources of AE during flight were:

(a) crack growth (including associated processes such as dislocation motion and brittle particle fracture);

(b) crack surface rubbing;

(c) fastener movement (within fastener holes); and

(d) movement between attachments to the component.

While for the majority of the flight trials the isolation zone was set to monitor the region containing holes 20F and 20R (see Figure 1), it was altered [1] for a short period to monitor an area of the component containing no holes or cracks, to test the effectiveness of the isolation method. During the period for which the isolation zone was so altered no 'Valid AE' were detected. This evidence suggests that the 'Valid AE' recorded during both the in-flight and laboratory portions of the overall test program, probably came from within the isolation zone. For the purpose of the following discussion it will be assumed that the isolation zone method used in this test program was operating effectively, and therefore all 'Valid' AE recorded was generated within the monitored zone.

As stated in section 5, no evidence of fretting/rubbing could be found in (or at) either of the holes within the isolation zone (i.e. 20F and 20R). All magnetic rubber plugs of holes 20F & 20R taken during the flight phase of the test were examined closely for any evidence of fretting/rubbing damage to the hole - no such evidence was seen. Indeed the only regions containing heavy rubbing/fretting damage to the centre section member were outside the extent of the isolation zone (Figure 6).

7. POSSIBLE SOURCES OF AE DURING LABORATORY TESTING

In the laboratory test, all fasteners and attachments were absent and the only possible AE sources were considered to be those associated with crack growth. It is considered unlikely that a significant number of the AE recorded in the laboratory by the aircraft system originated in the test rig or its connections to the test component, particularly in view of the noise attenuation measures employed. The results obtained using the AET 3000-LOCATOR in parallel with the aircraft system tend to support this view.

8. DISCUSSION AND CONCLUDING REMARKS

The results of the in-flight tests had shown an approximately linear relationship between the cumulative VALID AE counts and the total surface length of the cracks in the monitored zone. The objective of the laboratory tests, as indicated earlier, was to continue fatigue crack propagation under more controlled conditions in an attempt to identify the source of the AE recorded during the flight trials, and to compare results obtained from different equipment (AET 3000-LOCATOR).

Considering the extent of crack growth (2.53 mm increase in surface length) in the first half of the laboratory test, unexpectedly few emissions from the isolation zone were recorded by each of the independently operating monitoring systems (relative to the number of emissions recorded for a similar amount of cracking during the in-flight phase of testing; see Figure 4 and Table 1). For example, during the first half of the laboratory test a total of 1278 emissions were recorded by the aircraft system for the stated increase in crack surface length of 2.53 mm, while during the flight trials, for crack extension from 1.83 mm to 3.76 mm (i.e. 1.93 mm surface growth), 6801 emissions were recorded. Clearly this represents a marked decrease in the valid AE activity in the change from the in-flight to the laboratory phases of the test program.

The difference in the rates of AE counts with increasing crack surface length between the in-flight condition and the first half of the laboratory testing appears quite marked and is the major issue emerging from the present study. The load spectrum to which the component was subjected was

considered representative of the loading seen in service during operation of the aircraft. However, testing in the laboratory was purely axial (any bending being entirely restricted by the centre support assembly, see Figure 3) whereas when the boom is fitted in the aircraft, some bending occurs. Thus, loading of the monitored holes could ha e been different in the laboratory test. There appears no reason to suspect malfunction of the equipment and it will be assumed, for the purposes of the following discussion, that the results are substantially correct.

Firstly, it is important to examine whether or not a definite sharp transition in the AE vs crack length relationship exists at the change from in-flight to laboratory testing. Figure 7 shows that a smooth curve can be drawn through most, but not all, of the plotted data points; only the last in-flight point and one earlier point are displaced from the curve. These two displacements may or may not be significant; if they are not, then no sudden transition would be indicated, although the reason for the relationship exhibiting such a pronounced decrease in rate with increasing crack length would still be unexplained. It is to be emphasised that crack depth or crack area measurements, as opposed to crack surface length extension, would be more meaningful parameters to use for analysis of these data, particularly in view of the likelihood that all loading modes were not represented in the laboratory test configuration.

If a sharp transition does exist, then it must reflect some significant difference between conditions prevailing in flight and those in the laboratory. The most obvious difference is the attachment by fasteners of the spar boom to other components during the in-flight monitoring. Specifically, components which are attached to the spar boom when installed in the aircraft include: air-conditioning and engine inlet ducting brackets, shear plates and magnesium spacers forming an integral part of the wing carry-through structure, and various bolts, nuts, screws and washers. No fasteners or other attachments were present within, or near, the isolation zone during the laboratory tests. The transition to a lower rate of AE counts with increasing surface crack length at the start of the laboratory tests could therefore be explained by:

(i) fretting, or relative movement of components, which could have been generating AE above the threshold level during flight and been recorded as VALID AE. Thereby the AE count rate could have been augmented beyond that due to crack growth alone;

(ii) Stressing at the critical fastener holes being considerably different from what it had been in flight (e.g. the removal of the fasteners could allow greater change in the shape of the unsupported holes under load and encourage the cracks to grow along the surface rather than to increase in depth) so that fewer VALID AE counts would result for a given increase in crack surface length; and

(iii) Additionally, applic. tion of the 'marker' loads sequence at constant amplitude could have resulted in crack retardation with a similar effect on AE generation (if indeed crack growth is the actual source of AE). It should be noted however that an examination of crack growth data subsequent to the 'marker' loads (Table 1) does not support this view. Application of the 'marker loads may also have changed substantially any crack face rubbing characteristics which could well have been important in generating AE.

On the evidence available at present it is not possible to evaluate the above or other explanations; the cause of the rapid increase in AE rate in the second part of the laboratory test cannot yet be determined.

Regardless of which (if any) of the above explanations are found to be correct, the in-flight monitoring part of this test program has demonstrated that, with a suitably designed monitoring system, AE related to defects growing under fatigue loading can be detected in an aircraft during flight.

Fractographic analysis of the cracks from holes 20F and 20R has been completed, and a more detailed analysis of the in-flight data is being undertaken. Initial results have shown that the rate of Valid AE decreased for all flight regimes over the period of the flight trials (Figure 8); the use of AE rate as opposed to simple counts summation, when plotted against crack depth or crack area increase, may provide a more accurate representation of the flight trials data. A further report detailing these aspects of this project is in preparation.

9. ACKNOWLEDGMENTS

The author acknowledges helpful discussions with Mr L.M. Bland, Mr I.G. Scott and Dr F.P. Bullen of Aircraft Materials Division, ARL during the preparation of this paper. The equipment for the MACCHI programme was designed and constructed by Battelle Northwest Laboratories of Washington, U.S., under contract to Defence Advanced Research Projects Agency.

The assistance and the co-operation of ARL Structures Division staff is gratefully acknowledged.

The support and co-operation of the Royal Australian Air Force in various aspects of this programme is also acknowledged.

REFERENCES

(1) Martin, G.G., Mechanics of Non-Destructive Testing (Plenum Press), 1980, p343.

(2) Scott, I.G., Proc. 13th Symp. on NDE, San Antonio, 1981, p 210

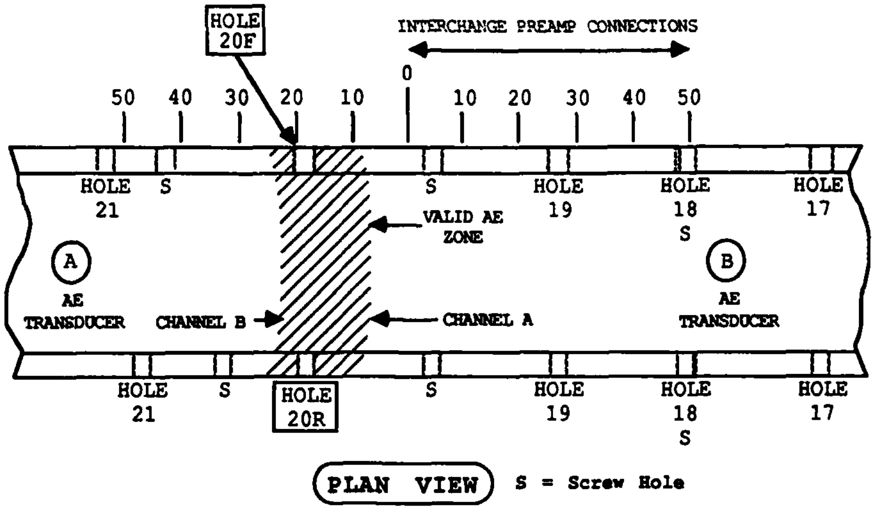
(3) Scott, I.G., Scala, C.M., Cousland, S.McK. and Rose, L.R.F, Metals Forum, 1982, 5, p 167.

(4) Parker, R.G., and Perry, F.C., Aeronautical Research Laboratories, Structures Note 481, 1982.

TABLE 1

In-Flight and Laboratory Test Results (Summarised)

REGIME	NUMBER OF FLIGHTS (PROGRAMMES)		INCREASE IN CRACK SURFACE LENGTH (mm)	VALID AE COUNTS		AE PER UNIT INCREASE IN CRACK LENGTH
				INCREMENT	TOTAL	
FLIGHT	838 (ACTUAL)		0.16 to 1.42 (1.26)	7312	7312	5800
			1.42 to 1.83 (0.41)	2231	9543	4340
			1.83 to 2.70 (0.87)	2766	12309	3180
			2.70 to 3.76 (1.06)	4035	16344	3800
LABORATORY	'MARKER' LOADS		3.76 to 4.01 (0.25)	18	16362	72
	900 (SIMUL ATED)	PROG 1	4.01 to 4.50 (0.49)	52	16414	106
		PROGS 2-3	4.50 to 5.5. (0.49)	377	16791	360
	900 (SIMUL ATED)	PROGS 4-5	5.55 to 6.29 (0.74)	831	17622	1120
		PROGS 6-9	6.29 to 7.33 (1.04)	20743	38365	~20000



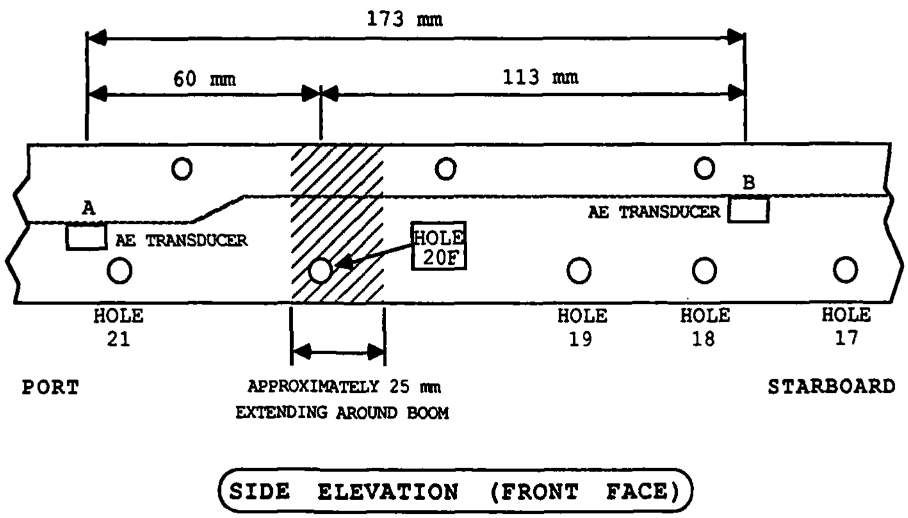


Figure 1. Location of AE transducers and isolation zone on MACCHI wing centre section assembly tension boom. Cracking was known to be occurring in holes 20F and 20R.

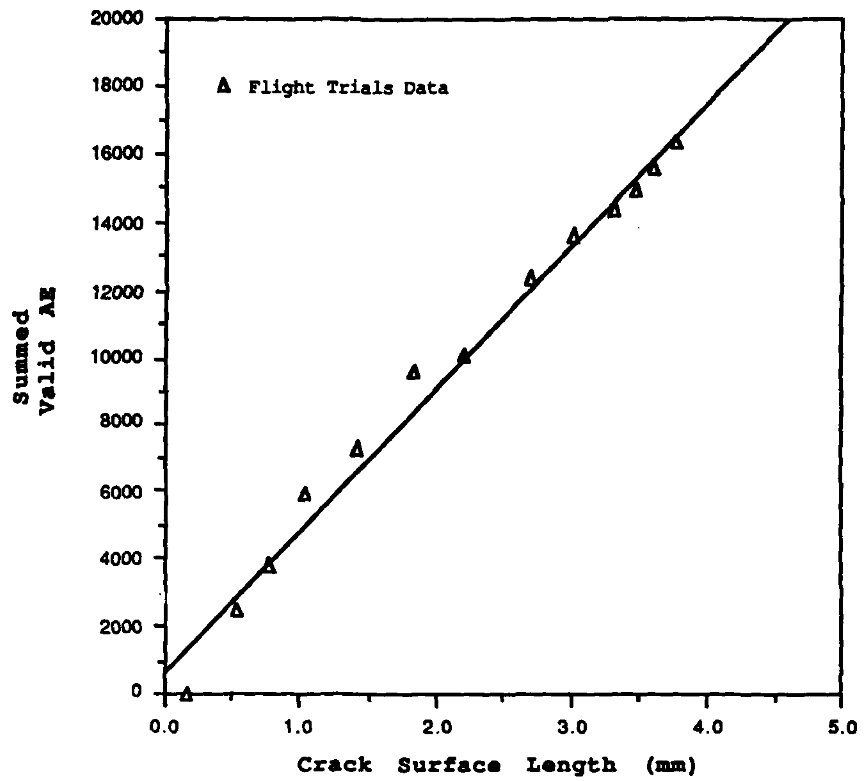


FIGURE 2. Summed VALID AE plotted against crack surface length from the magnetic rubber inspection (MRI) replicas of cracks in hole 20, front and rear face. It is to be noted, in comparing Figure 2 with the corresponding plots in references 1,2 and 3, that a more comprehensive correction (based on the total flight data available) has been applied in the present analysis concerning the 210 flights, out of 838, for which AE counts from the monitored zone were not available. The solid line represents the 'line of best fit' with a correlation coefficient of 0.99

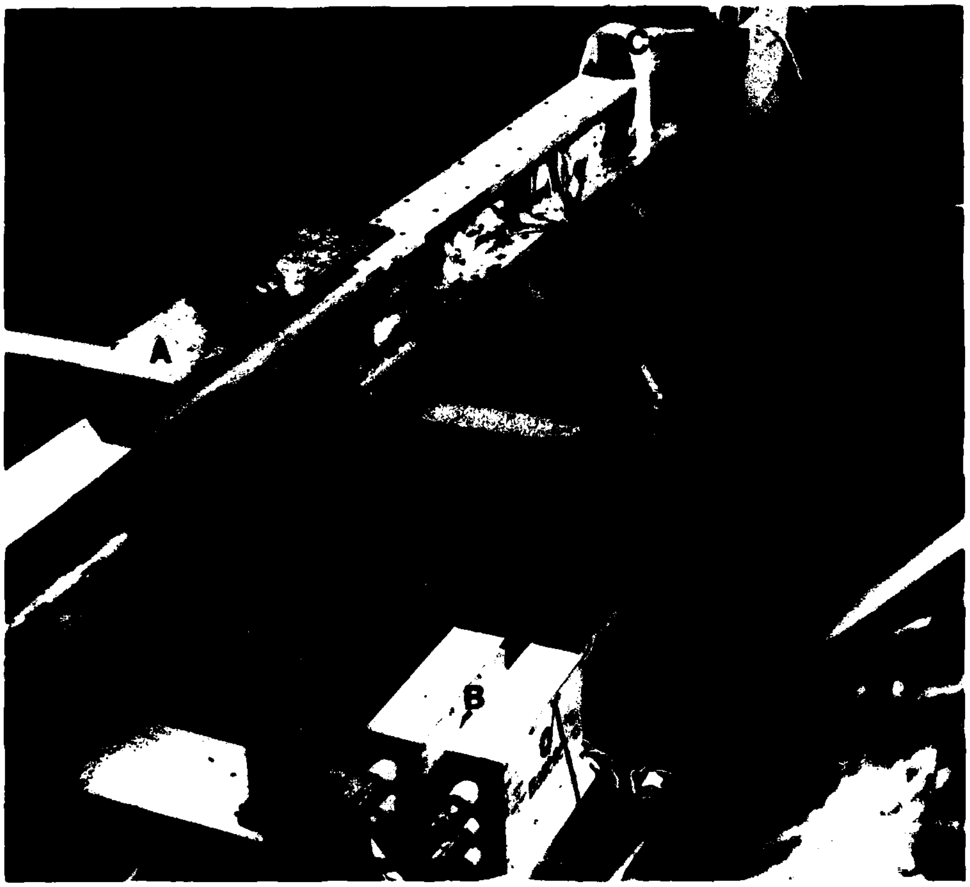


FIGURE 3. Detail of tension boom in test rig showing centre support framework (A); aircraft AE system pre-amplifiers (B); aircraft AE system transducers (T) and one of the two AET 3000-LOCATOR system transducers (C). Note that all cabling was taped clear of the component during testing to prevent injection of spurious AE signals.

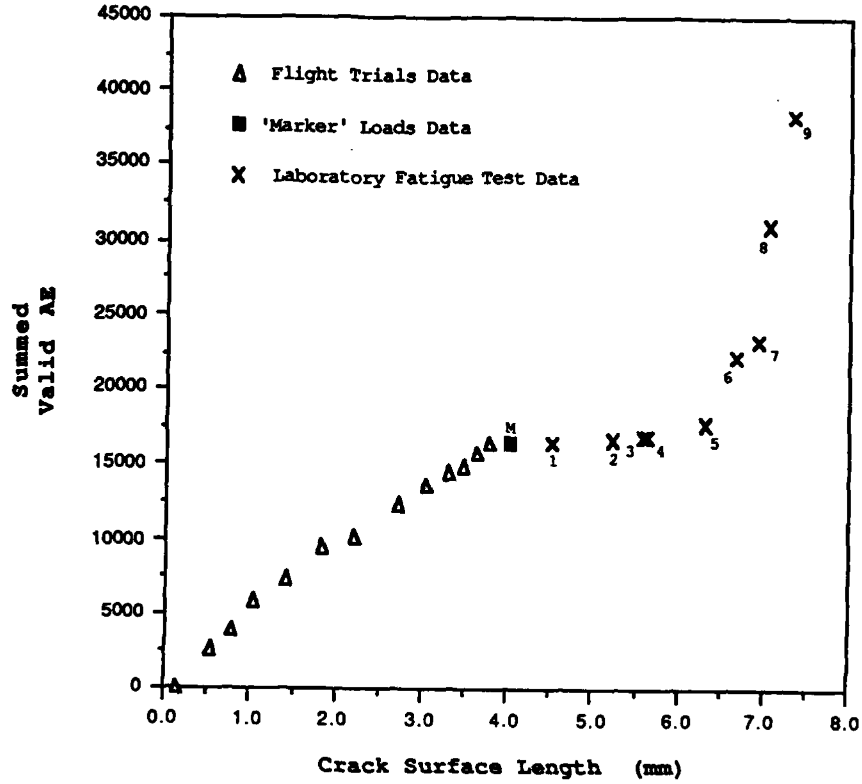
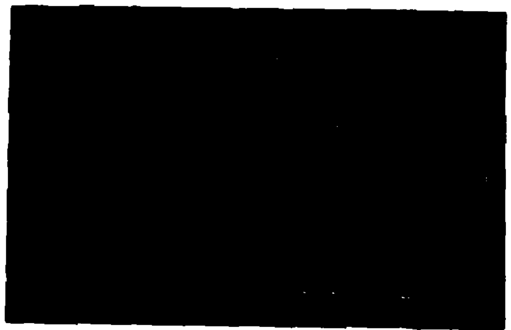


FIGURE 4. Summed AE versus Crack Surface Length over the duration of the flight trials, together with data from the laboratory fatigue test. The first point (M) of the laboratory data corresponds to the application of the 'marker' loads sequence.

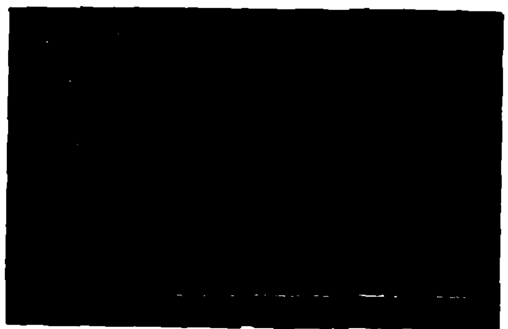
PROGRAM 2.

Hole 20

PROGRAM 5.



PROGRAM 7.



PROGRAM 9.

FIGURE 5. Photographic records of the CRO screen of the AET 3000-LOCATOR monitoring system at the end of the laboratory testing loading programs 2,5,7 and 9. (The horizontal length of the screen record corresponds to the 975 mm separation of the monitoring system transducers mounted at each end of the component. The longitudinal position of holes 20F and 20R is marked by the arrow in each photograph. The vertical scale corresponds to the number of counts recorded for each longitudinal position during the nominated programs.

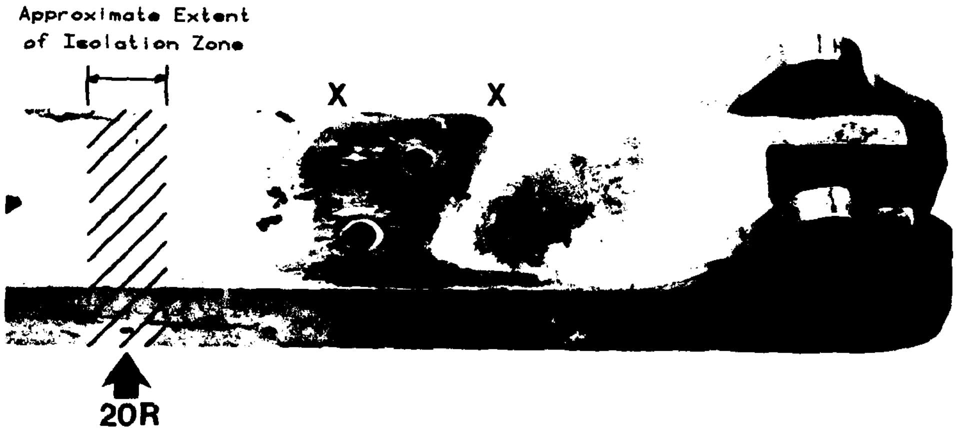


FIGURE 6. Starboard end of centre section component showing imprint (X–X) of contacting structural member produced by rubbing/fretting action during operation of the aircraft. The approximate location of transducer A (on under-surface, refer Figure 1) is shown together with the position of hole 20R.

MAGNIFICATION: 0.6 (approx.)

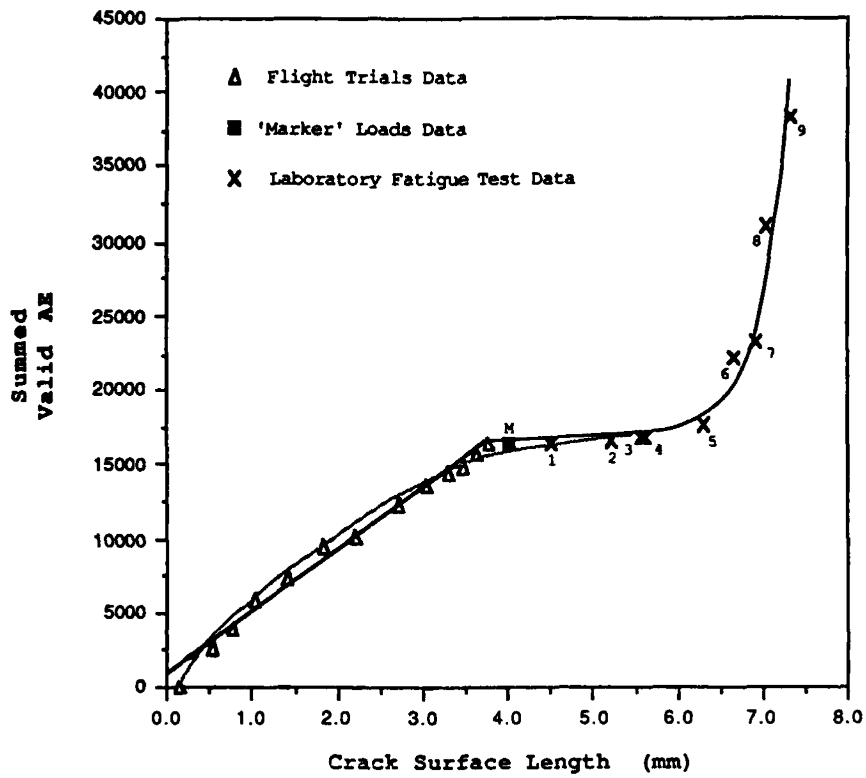


FIGURE 7. Two possible interpretations of the experimental data are shown. In the first case (solid line) a linear relationship is defined for the flight regime, with a marked discontinuity upon transition to the laboratory test conditions. In the second case a curvilinear relationship is defined, which remains continuous through the change from the conditions of the flight trials to those of the laboratory test.

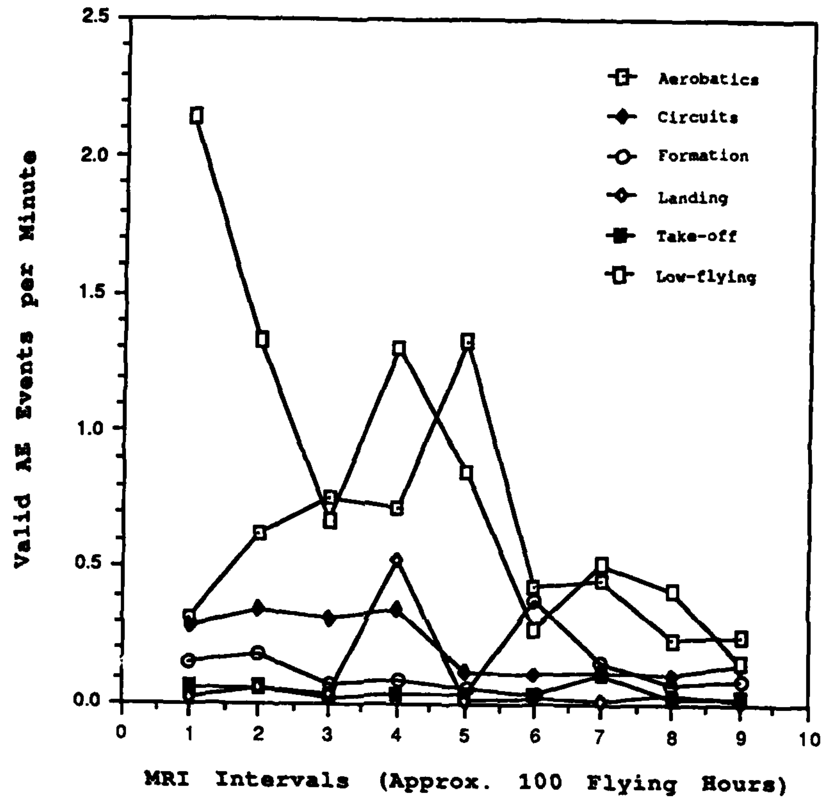


Figure 8. VALID AE event rate (counts per minute) plotted over each magnetic rubber inspection interval. Note that, in general, each flight regime exhibits a decrease in the rate of AE over the period of in-flight monitoring.

DISTRIBUTION

AUSTRALIA

Department of Defence

Defence Central
- Chief Defence Scientist
- Deputy Chief Defence Scientist (shared copy)
- Superintendent, Science and Program Administration (shared copy)
- Controller, External Relations, Projects and
  Analytical Studies (shared copy)
- Counsellor, Defence Science (London) (Doc Data Sheet Only)
- Counsellor, Defence Science (Washington) (Doc Data Sheet Only)
- SA to the Thailand Military R and D Centre (Doc Data sheet only)
- SA to the D R C (Kuala Lumpur) (Doc Data sheet only)
- OIC TRS, Defence Central Library
- Document Exchange Centre, DISB (18 copies)
- Joint Intelligence Organisation
- Librarian H Block, Victoria Barracks, Melbourne
- Director General - Army Development (NSO) (4 copies)
- Defence Industry and Materiel Policy, FAS

Aeronautical Research Laboratories
- Director
- Library
- Superintendent - Aircraft Materials
- Divisional File - Aircraft Materials
- Author : S.R. Lamb
  - G.Clark
  - D.R.Arnott
  - C.M.Scala
  - S.J.Bowles

Materials Research Laboratories
- Director/Library

Defence Research Centre
- Library

Air Force Office
- Air Force Scientific Adviser
- Aircraft Research and Development Unit
  - Scientific Flight Group
  - Library
- Technical Division Library
- Director General Aircraft Engineering - Air Force
- Director General Operations Requirements - Air Force
- HQ Operational Command (SMAINTSO)
- HQ Support Command (SLENGO)
- RAAF College, Point Cook

SPARES (10 copies)
TOTAL (57 copies)

Department of Defence

DOCUMENT CONTROL DATA

1.a. AR No	1.b. Establishment No	2. Document Date	3. Task No
AR-004-531	ARL-MAT-TM-393	APRIL 1987	AIR80/043

4. Title	5. Security a. document	6. No Pages
LABORATORY STUDIES RELATED TO IN-FLIGHT ACOUSTIC EMISSION MONITORING	UNCLASSIFIED	18
	b. title: U c. abstract: U	7. No Refs: 4

8. Author(s)

S.R.Lamb

9. Downgrading Instructions

-

10. Corporate Author and Address

Aeronautical Research Laboratories
P.O. Box 4331,
MELBOURNE, VIC. 3001

11. Authority (as appropriate)
a.Sponsor b.Security c.Downgrading d.Approval

-

12. Secondary Distribution (of this document)

Approved for Public Release

Overseas enquirers outside stated limitations should be referred through ASDIS, Defence Information Services Branch, Department of Defence, Campbell Park, CANBERRA ACT 2601

13.a. This document may be ANNOUNCED in catalogues and awareness services available to ...

No Limitations

13.b. Citation for other purposes (ie casual announcement) may be (select) unrestricted~~(or) as for 13 a.~~

14. Descriptors

Acoustic Emissions
Crack Propalation
Nondestructive Testing

15. ~~COSATI Group~~

DRDA
Subject
Categories
0094J

16. Abstract

Programmed load testing of a structural member from a MACCHI aircraft was undertaken in an attempt to determine the source of the acoustic emission previously recorded during in-flight monitoring of the same component and to compare results obtained from different equipment. Although crack growth during laboratory testing appeared similar to in-flight crack growth, the pattern of recorded AE was markedly different. The laboratory tests are described and an explanation of the test results sought.

PF 65

This paper is to be used to record information which is required by the Establishment for its own use but which will not be added to the DISTIS data base unless specifically requested.

16. Abstract (contd)

17. Imprint

Aeronautical Research Laboratories, Melbourne

18. Document Series and Number	19. Cost Code	20. Type of Report and Period Covered
Aircraft Materials Technical Memorandum 393	311256	

21. Computer Programs Used

22. Establishment File Ref(s)

M1/8/469

END

DATE
FILMED

1 8